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AIR BREATHING JET ENGINES - BACKGROUND
These notes form a reasonably comprehensive treatment of the principal air-breathing jet engines available for missile propulsion. Following a brief introduction in section 1, section 2 contains an account of the history of these engines in missile propulsion from about 1940 onwards. Section 3 is devoted to intakes, which are vitally important components of air-breathing engines. The emphasis is on supersonic intakes and their performance characteristics; however, part of this section deals briefly with subsonic intakes, reflecting the increasing importance of missiles in this category. Sections 4 and 5 respectively deal with ramjets (and to some extent ramrockets) and turbojets/turbofans. In each section, consideration is given to the thermodynamic cycle, the method of estimating design point performance, the choice of the “best” cycle for a given mission, the off-design engine performance and their physical layouts.
NOTATION
A area
C Coefficient (ie lift, drag etc.)
D Drag
F thrust
C function of M and y in adiabatic flow
g gravitational acceleration
h specific enthalpy
h_o specific total enthalpy
H_c calorific value
K,k constants
C_p specific heat at constant pressure
C_v specific heat at constant volume
L lift
M Mach number
N function of M and y in diabatic flow
p static pressure
p_o total pressure
m mass flow
q, f fuel/air ratio
R gas constant
r pressure ratio
s.c. specific consumption
T static temperature
T_o total temperature
u,v velocity
X,Y,Z functions of y and N
γ ratio of specific heats
ρ density
Suffices

a  ambient of value for air
b  value behind shock external to engine
d  drag
d  value in diffuser
e  value for nozzle
f  value for fuel
H  based on enthalpy
i  value at inlet
K.E. based on kinetic energy
L  lift
n  value of nozzle
max  maximum value
p  based on pressure
T  thrust
t  value at throat
0,1,2.. values at successive planes in engine
1. INTRODUCTION

It is well known that all guided missiles (with the possible exception of torpedoes) employ jet propulsion, of one form or another. Additionally, it can be accurately stated that the predominant form of jet propulsion is the solid-propellant rocket motor. This state of affairs has existed for about 50 years and will certainly continue into the foreseeable future. It is therefore impossible at present to envisage the solid-propellant motor being displaced from its wide-spread employment in missile systems of nearly all categories – strategic ballistic, tactical ballistic, high-level and low-level air defence, anti-ship, anti-tank, air-to-air and so on.

However, over these 50 years or so, there have been deployed missiles which used, wholly or partially, jet propulsion by an air-breathing engine. Compared with rocket-propelled missiles, their numbers have always been small and at times there perhaps seemed to be few avenues open to the air-breathing jet engine and so interest in the air-breather tended to fluctuate. Now we have entered the 21st century, it is fair to say that the air-breather is enjoying a measure of resurgence in missile propulsion. It is therefore appropriate to outline the historical development of air-breathing engines in missiles, to examine the reasons for their current resurgence, to consider the performance characteristics of the key types of engine available and to discuss the particular present-day mechanical features special to air-breathers for missile propulsion.
2. HISTORICAL BACKGROUND

In this section nearly all of the missiles mentioned were, are or will be, in service with armed forces. No doubt there were others of which the author is unaware. Also it is certain that there have been, over these 50 years, many programmes of research and development on systems which did not reach completion or were, in some way or other, unsatisfactory and so did not enter the armouries of the world. Equally, it is certain that under development now are systems that have yet to go into service. Such missiles may not be known to the public, even in the Western World. Consequently, the missiles discussed in this section cannot constitute a complete collection.

2.1 The pulsejet

The first missile to be considered is, of course, the WW2 German V1 “Flying Bomb”. This missile, with a rudimentary form of inertial guidance, incorporated a very unusual type of air-breathing jet propulsion unit, known as a pulsejet.

![Pulsejet diagram]

Figure 2.1 – Pulse-jet as used on V1 Flying Bomb

This type of engine is particularly interesting in that it is a non-steady-flow device. Rocket motors, ramjets and turbojets are essentially steady-flow devices (this feature makes them relatively simple to analyse) whereas the pulsejet is in a completely different category known as intermittent jets. The principle of operation was discussed in the “Introduction to Propulsion” lectures but some features of the V1 engine will be mentioned further now.
Its main virtue was its cheapness; other points in its favour were no doubt its lightness, lack of moving parts, ease of production and use of a conventional hydrocarbon fuel. Contrasting features were its poor efficiency (i.e. high specific fuel consumption), its low reliability and the adverse effect its pulsing thrust had on whole missile airframe. In particular, the mean thrust of the unit was about 2500 N but the peak thrust was about 10 times the mean. However, the unit could accelerate the flying bomb from its launch velocity of about 400 km/h to about 625 km/h and sustain it at that speed on a typical flight from the French coast to Greater London.

After the employment of the pulsejet in the V1, work was undertaken in various Western countries (and doubtless in the then Soviet Union also) to properly analyse and model the operating characteristics of the intermittent jet, of which there are a number of variants. The pulsing nature of its operation and the consequent pulsing thrust characteristic which imposes an undesirable dynamic loading on the missile, together with its inherently high fuel consumption, are probably the features which account for its non-appearance since WW2 as a missile propulsion unit. An exception is the Swedish Robot 315, a ship-borne bombardment missile (in service about 1958) which apparently employed a pulse jet.

2.2 The ramjet and ramrocket (or ducted rocket)

Of greater importance since WW2 is the ramjet (and to a lesser extent the ramrocket), which has been employed in a number of missile systems and which, in the 1970’s and 80’s, was the subject of revived interest, at least in terms of R and D.

One of the earliest and most interesting projects involving the ramjet was the Navajo missile system (which was to be operated by SAC). In about 1958/60 the system was under development alongside the first generation Atlas ICBM. The missile, which was to have an intercontinental range, was to be surface-launched (presumably by liquid-propellant rocket engines, at least in its original version) into an aerodynamic trajectory in which it was sustained at Mach 3 at around 75,000 – 90,000 ft altitude by a pair of wing-mounted ramjets, each having a diameter of about 1.2 m and burning kerosene. It is noted in Flight magazine’s Missile Systems of 1960 that the ramjets were “under development”. Information on other aspects of the Navajo missile is scant, but the size of the twin propulsion units does give an indication of the likely overall size of the missile (compare with the Snark - see later). It is understood that the programme was cancelled around 1960.
At the same time the vertically-launched air-defence missile system *Bomarc* went into service. The Bomarc (Boeing Michigan Aeronautical Research Centre) missile, sometimes described as a “pilotless interceptor”, had a launch weight of about 7000 kg and certainly in its original version was launched by a nitric acid/kerosene liquid-propellant rocket engine. Later versions used a solid-propellant motor. After boost, the missile cruised at Mach 2.8 out to a maximum range of
about 350 km at a height of about 70,000 ft under the thrust of 2 Marquardt podded, kerosene-
burning ramjets of 0.71 m diameter. It is thought that the missile was taken out of service around
1970. There have been suggestions that it has been since used as a supersonic target vehicle.
Also around 1960 the US Navy put into service a ship-launched, surface-to-air missile called
*Tabs* (a mythical, bull-headed giant) with a launch weight of rather less than half that of
Bomarc. Boost was by a single tandem solid-propellant rocket motor but, in contrast to the
Bomarc system, sustain propulsion was provided by a single integral ramjet whose air was
supplied via a transfer duct from the centre-body axisymmetric intake positioned at the nose of
the missile. Guidance was by beam riding with SA homing for the terminal phase. Boost burnout
occurred at Mach 2.5, necessitating acceleration up to Mach 3 by the ramjet. Tabs is said to have
had a range greater than 120 km and a ceiling of about 85,000 ft. The combustor, developed by
APL Johns Hopkins University, used a mixture of kerosene and naphtha. The system is no
longer in service.

It was in 1952 that the first Bristol ramjet was flown, paving the way for development of the
podded ramjet in the UK. It is almost wholly in what is now the Bristol division of Rolls-Royce
that the expertise in ramjet development and production resided. The *Thor* ramjet, two of which
provide the cruise propulsion for the *Bloodhound* missile, was the outcome of the ramjet
programme. Around 1960 the UK Bloodhound Mk 1 surface-to-air missile system went into
service, the Mk 2 version following in 1964. Sustain propulsion up to a maximum range of
about 200 km was provided by two Thor kerosene-burning podded ramjets, stub-mounted on
either side of the main missile body. The launch weight (including 4 wrap-around booster
motors using solid plastic propellant) was about 2000 kg, considerably less than the Tabs 2.
Maximum speed and height were about Mach 2.6 and 60,000 ft respectively. This system
remained in service with the RAF in the UK until the late 80’s.

Technical intelligence estimates are that at around 1964 (the time that Bloodhound Mk 2 went
into service) the Russian *SA-4* missile became operational. With a launch weight of 1800 kg, this
missile employed four wrap-around booster motors and cruise propulsion by an integral ramjet
with an axisymmetric nose intake. It is thought that the ramjet was kerosene-fuelled giving a
range of more than 70 km at around Mach 2.5 up to an altitude of 80,000 ft.

The ramjet configurations mentioned so far have all been what could be described as
conventional in that they use:

a. liquid hydrocarbon fuel;

b. axisymmetric podded or integral designs;
c. separate booster motors.

The Russians broke new ground (taking the West by surprise) in “ramjet” propulsion in in-service systems when at around 1967 they deployed the **SA-6 ramrocket** (or **ducted rocket**)-propelled missile, which was to be used by the Arabs against Israeli-flown American-made aircraft in 1973. The propulsion system in this missile employs several features which were then novel:

a. a solid-propellant gas generator: a very fuel-rich propellant is burned in the style of a solid-propellant motor to provide fuel-rich gas for the main combustor downstream;

b. four 1/2-axisymmetric intakes: positioned well rearwards of the missile nose to feed air to the combustor;

c. an integral booster motor: the propellant grain (i.e. charge) temporarily installed in the ramjet combustor, which has its air intakes temporarily blocked off, burns out through its own ejectable nozzle, positioned within the ramjet nozzle.

These features offer a much improved compactness of the whole missile and a much simplified nose section free of intake hardware. This missile is believed to be in service still, possibly in the Middle East only. Countering these benefits are several disadvantages which will be mentioned later in these notes.

Of the operational missiles in the West, the **Sea Dart** is still in service with the RN in the UK. It is primarily an area-defence surface-to-air semi-active homing missile with a range of around 80 km and a maximum altitude of about 80,000 ft. It also has a surface-to-surface role. Its launch weight (550 kg) is the same as that of the SA-6. Boost from rest is by a tandem solid-propellant motor employing a very energetic CMDB propellant whilst cruise propulsion is provided by a single integral ramjet whose axisymmetric intake is at the nose of the missile. As in Tabs, a transfer duct delivers the intake air to the combustor/nozzle assembly which is necessarily at the rear of the missile. Normal aviation grade kerosene is the fuel and speeds of up to Mach 2.8 are possible.
The most recently fielded ramjet-propelled missile in the West is the Aerospatiale ASMP (Air-Sol Moyenne Portée) which became operational in 1986. This air-launched missile can deliver a nuclear warhead at up to 250 km flying at Mach 3 at high level or up to 80 km at Mach 2 in the sea-skimming role. Propulsion is by an integral solid-propellant booster (5 s duration) then kerosene-burning ramjet which has two side-mounted 2-D intakes.

A characteristic common to all the missiles discussed so far is their supersonic cruising velocity. They are all designed to cruise in excess of Mach 2, some just beyond Mach 3, at high altitude. Ignoring any operational advantages resulting from these high speeds, it is of course fundamental to the ramjet mode of operation that a high speed is achieved. This is a direct consequence of the fact that a ramjet or ramrocket is a compressorless, steady-flow jet propulsion device. More will be said about the necessity of a very high forward speed later, but it contrasts with the missiles discussed in section 2.3 (all of which employ turbojets or turbofans) among which only a few are supersonic.

Current ramjet and ramrocket programmes, which may or may not eventually feature in fielded missile systems, include:

a. FRANCE: ONERA; rustique ramrocket, self-modulating type (subsonic gas generator orifice), proposed by MATRA for possible SAMIAAM applications.

b. USA: USAF; Variable-flow ducted rocket (VFDR); valve at gas generator exit sonic orifice(s) for throttling; low smoke, low metal propellant; possible FMRAAM application.

c. GERMANY: DASA; variable-flow ramrocket; same principle as in (b) above; boron-loaded fuel for gas generator; possible FMIRAAIVI application, viz A3M.

d. FRANCE: Aerospatiale; liquid hydrocarbon fuel, based on existing ASMIP (see earlier); for Aerospatiale/DASA ANNG (also known as ANF) supersonic anti-ship missile (ANS abandoned).

e. UK/USA/FRANCE: Matra-BAE Systems/Boeing Meteor: Liquid fueled ramjet with a range of 80 km. Likely entry service date, 2010.

f. USA: Hydrocarbon and boron solid-fuel ramjet; possible application to short-range air to-surface missiles.

2.3 The turbojet and turbofan

Until about 1980, the turbojet and turbofan were known almost exclusively as aircraft propulsion units. Consequently, up to then the relatively few gas-turbine jet engines employed in
missile propulsion were aircraft engines, i.e. man-rated, incorporating a few minor modifications reflecting the much more restricted flight envelope of a typical “cruise” missile. This situation has changed in response to the particular propulsion requirements of today’s relatively small subsonic cruise missiles, namely the need to match engines precisely to their job in terms of volume, weight, specific fuel consumption, reliability and cost. To meet these requirements in what can loosely be described as a “small” engine (up to perhaps 300 mm maximum diameter), is not a simple task.

Figure 2.4 – Typical Turbojet and Turbofan Schematics

Thus the gas-turbine jet engine, from the 50’s to the early 70’s, as a match to the propulsion requirements of a missile, was in a situation quite different from that of the ramjet. The ramjet has nearly always been envisaged as and designed as an operationally short-life unit for a short-life application, the missile flight. On the other hand, the turbine engine (with one or two exceptions) has been designed with the intention of obtaining a “long” operational life, perhaps up to thousands of hours, as in engines for civil aircraft. The implications of this contrasting situation in terms of design, material selection, manufacturing techniques, propulsive performance and size are of considerable significance.

For over 20 years up to the 70’s there was deployed by the USA and USSR (and possibly others) a range of turbojet-propelled missiles of both tactical and strategic significance. One of the most interesting was the Snark bombardment missile built by Northrop. Developed at a similar time to the Navajo system, the Snark entered service with the SAC (around 1959), unlike the Navajo. With a range of 10,000 km, its capabilities were presumably similar to those of the Navajo missile, namely the ability to deliver a nuclear warhead over intercontinental range. However, the Snark was subsonic, being designed to cruise at about Mach 0.94 at a height of 60,000 ft. It was surface-launched with the aid of two jettisonable solid-propellant rocket boosters. Launch weight was 22,000 kg. Cruise thrust was provided by a single P&W J-57 turbojet engine
(delivering up to 58,000 N of static thrust at sea level) mounted under the rear fuselage in a pod. Today, this missile would be very vulnerable to air-defence missiles, but of course one must assess an attack system’s capabilities in conjunction with the then current defensive capabilities. Whilst the Snark was the only airborne in-service strategic missile with a range of the same order of magnitude as the ICBM, there were several other “long-range” missiles deployed by the USA (and the USSR) which could also be classified as strategic. For example, the North American Hound Dog air-launched missile with a maximum range of 800 km was in service in the 60’s and 70’s. Its cruise propulsion was provided by a P&W J-52 turbojet of 33,000 N static thrust at sea level. Inertially guided, it could deliver a nuclear warhead. It was the only supersonic turbojet-propelled missile of American origin to enter service; cruise speed was around Mach 1.6. In service, a missile was mounted under each wing of the B-52G. By all accounts the engine in each missile could be run to provide extra take-off thrust, a novel feature. Fuel consumed in the process was made up from the aircraft’s tanks. Not surprisingly, in order to obtain efficient take-off and Mach 1.6 performance, the missile engine used a moveable centre-body air intake and a moveable plug nozzle.

The USA also deployed around 1960 with the Air Force the subsonic Matador and later Mace missiles which were ground-launched bombardment missiles, the latter having a range of up to about 2200 km. Around the same period, the US Navy had an operational ship-launched missile, Regulus, with a subsonic range of up to 900 km. All three missiles were sustained during cruise by the Allison 1-33 turbojet, a man-rated engine.

All five missiles mentioned so far would probably be described as first generation cruise missiles, a category of missile that has received much publicity in recent years. In propulsion terms, they exemplify the principle outlined earlier that cruise missiles of that period employing turbojets used engines which were first developed for aircraft propulsion. This is hardly surprising since those missiles were of aeroplane size and configuration. Engines of approximately the right size were therefore available, or under development, and so the economic solution would be to incorporate straight-forward modifications to the existing engines to adapt them for missile propulsion. However, those engines were far removed from matching the missile’s designed-in expendability and so strictly speaking were not the best engines for the job. The contrast with today’s generation of cruise missile is very marked.

Over the same period, the former Soviet Union designed, developed and put into service various “large” cruise missiles employing turbojet propulsion, much as the USA did. Scrutiny of past editions of reference publications reveals that the Soviet Union deployed several different types
in the air-to-surface and ship-to-ship categories. Information is patchy but it’s sensible to assume that the types were similar in performance to the American systems.

There are one or two nations apart from the USA and the former Soviet Union which have developed and put into service turbojet-propelled missiles. The Italian/French Otomat ship-launched anti-ship missile (in its Mk 2 version a sea-skimmer with a range of 100 km +) cruises under the power of a Turbomeca Arbizon turbojet, an engine of conventional design. This system became operational with the Italian Navy in 1978. Some years ago the Royal Swedish Navy operated the RB 08A ship-to-ship missile which employed a Turbomeca Marbore turbojet. Current in-service missiles in this category are particularly interesting in that they are the first examples of missiles in the West employing engines which were conceived outside the “man-rated” concept in the classical approach to aero-engine design; the engines in this group were conceived within what can rather loosely be described as the UAV/missile concept. Some of the major characteristics of these missiles will now be examined, bearing in mind that there are several versions of Tomahawk, from the strategic land-attack type to tactical land and ship attack weapons.

The most notable feature of Harpoon, Sea Eagle, ALCM and Tomahawk is that they are considerably lighter than the systems of the 50’s and 60’s. They employ the turbojet or the turbofan of sea-level static thrust of approximately 300 kN. Thrusts of this magnitude required new engines and so truly expendable engines were designed in the USA and France (and in the Soviet Union also). In France, Microturbo are responsible for the Sea Eagle’s turbojet. In the USA, Teledyne produced the Harpoon’s turbojet and the Williams Corporation produced the turbofans for the ALCM (AGM-86) and the Tomahawk variants.

This outline of gas-turbine jet engine applications in missiles over the past 40 years or so indicates that, after a long period during which man-rated engines were used, mainly because the cruise weights of the missiles necessitated fairly large or large engines (which already existed for conventional aircraft propulsion), there is now a specific class of truly expendable small engines. This is mainly because both the tactical and theatre/strategic subsonic cruise missiles now in service are relatively small.

Current procurement efforts in the West featuring turbojet and turbofan propulsion are as follows:

a. FRANCE: Matra; Apache, Microturbo turbojet, about 180 km range.

b. FRANCE/UK: Matra BAe Dynamics; Storm Shadow (CASOM), air-launched derivative of Apache with same propulsion system, range > 350 km (SCALP virtually identical).
c. FRANCE: Matra; Apache derivatives: APTGD “Armement de Precision Tire a Grande Distance”.

d. USA: FOG-M applications, turbojet; very small turbojets (for target drones, decoys and maybe missiles) up to 100 mm maximum diameter and 5 kg mass. Example: Sundstrand TI-SO (50 lbf thrust at sea level), some silicon nitride components.

e. USA: Advanced Cruise Missile, maybe new developments for turbofan.
3. THE INTAKE

3.1 Introduction

The intake is of prime importance for all air-breathing missile propulsion systems. Its major function is to collect the atmospheric air (its working fluid) at free-stream Mach number, slow it down (probably involving a change of direction) and so compress it efficiently, i.e. as nearly as possible, reversibly. In this role the intake is performing an essential part of the engine cycle and its efficiency is directly reflected in the engine performance. In addition, the intake must present the air to the downstream component at suitable velocities and with an acceptable degree of uniformity of velocity and pressure under all flight conditions. This engine-intake compatibility is essential for successful operation of the missile within the desired flight envelope. Finally, the intake has to achieve all this with minimum external drag and minimum disturbance to the external flow around the missile.

At low speeds, the dynamic pressure is small and the static pressure rise obtainable by deceleration of the air (see Gas Dynamics Notes) is insignificant compared with engine cycle requirements. In the subsonic engine, the intake design therefore tends to be dominated by external flow considerations.

At supersonic speeds the pressure recovery is of great importance and may, as in the case of the ramjet, provide the whole of the engine cycle compression.

The following subsections are concerned mainly with the supersonic intake, however there is a brief description of the main features of the subsonic intake.

3.2 Definitions for performance of intakes.

Ideally the compression accomplished by an intake would be isentropic. This ideal is never achieved and the commonly used criteria of intake performance are therefore defined as follows. Let the suffixes a and d refer to free stream values and values at the diffuser exit respectively. The non-isentropic, i.e. irreversible process, can then be drawn on the temperature-entropy (T-s) graph in Figure 3.1.
The following criteria are defined:

(i) The pressure recovery $\eta_d$ is the ratio of mean total pressure at the diffuser exit to free stream total pressure, i.e.

$$\eta_d = \frac{p_{a_d}}{p_{a_o}} \quad (1)$$

(ii) The isentropic efficiency of the intake of the intake $\eta_i$ is defined as:

$$\eta_i = \frac{T_{a_d} - T_a}{T_{a_d} - T_a} \quad (2a)$$

Therefore

$$T_{a_d} - T_a = \eta_i \left( T_{a_d} - T_o \right) = \eta_i \frac{u_{a_d}^2}{2c_p} \quad (2b)$$

and

Figure 3.1 – T-s diagram showing intake process
\[
\frac{p_{o_x}}{p_a} = \left( \frac{T_{o_x}}{T_a} \right)^{\frac{\gamma}{\gamma-1}} = \left[ 1 + \frac{T_{o_x} - T_a}{T_a} \right]^{\frac{\gamma}{\gamma-1}} \\
= \left[ 1 + \eta_i \frac{u_a^2}{2c_p T_a} \right]^{\frac{\gamma}{\gamma-1}} = \left[ 1 + \eta_i \frac{\gamma - 1}{2} M_a^2 \right]^{\frac{\gamma}{\gamma-1}}
\]

The former criterion is used when pressure recovery is of great significance in the engine cycle, e.g. in the case of a ramjet, while the latter criterion is used when pressure recovery is of less significance in the engine cycle and minimisation of energy degradation (or degree of irreversibility) is of more importance, e.g. in the intake of a subsonic turbojet.

Also note the simple relationship between the two criteria:

\[
\frac{p_{o_x}}{p_a} = \left[ 1 + \eta_i \frac{\gamma - 1}{2} M_a^2 \right]^{\frac{\gamma}{\gamma-1}}
\]

This emphasises the fact that pressure recovery is not an efficiency, being a function of free stream Mach number.

3.3 The Subsonic Intake

The most common type of subsonic intake is the pitot intake which consists of a simple forward facing entry hole with radiused cowl lips. The three major types of pitot intakes are shown in Figure 3.2. These are podded intakes, integrated intakes and flush intakes. The podded intake is uncommon in missile propulsion and is used mainly in subsonic transport aircraft. The integrated intake (Tomahawk) and flush intake (Harpoon) are more commonly found since they can be more readily accommodated into missile airframes.
Typical flow characteristics of pitot intakes are illustrated in Figure 3.3 for four flow conditions. At high speed cruise (a) (M = 0.85) the entry streamtube will be smaller than the intake area since the intake will diffuse the flow and hence lower the intake velocity with a small resultant rise in pressure (15%). In climb (b) the freestream velocity will be lower than the intake velocity due to the high mass flow rates required. This will result in a larger entry streamtube area than the intake area. In ground running (c) there will be no effective freestream velocity which results in a large induced flow capture area causing the streamlines to converge into the intake area. At
top speed \((d)\) (higher than cruise \([M = 0.95]\)), the high pressure gradient on the intake lip can cause separation and an unstable flow into the intake.

Figure 3.4 – Performance of subsonic intakes at static and dynamic conditions

Figure 3.4 shows typical pressure recovery values for subsonic intakes based on various intake shapes. It can be seen that the more highly radiused the intake shape (i.e. greater contraction ratios), the better the pressure recovery values for given freestream Mach numbers. This is due to the greater radii reducing the pressure gradient and hence losses into the intake. The main concern with subsonic intake design is to avoid areas of high Mach number in the lip. This is usually done by choosing the correct aerofoil profile in the entrance region. As Figure 3.4 shows, a good subsonic intake should achieve a pressure recovery of over 90%. Therefore much of the design work with subsonic intakes involves the reduction of external drag.

3.4 The Supersonic Intake

Several important parameters and operating modes must be considered when analysing supersonic intakes.

(a) Modes of Operation of the Supersonic Intake

The operating range of all supersonic intakes can be divided into 3 distinct modes, thus:-

(i) CRITICAL condition — the point at which maximum mass flow is reached at
any speed;

(ii) SUBCRITICAL condition — all operating conditions in which the mass flow is less than that at the critical condition;

(iii) SUPERCRITICAL condition — all operating conditions at constant i.e. maximum, mass flow except the precise critical condition.

(b) Mass Flow Ratio ($\varepsilon_d$)

![Schematic to illustrate mass flow ratio](image)

Figure 3.5 – Schematic to illustrate mass flow ratio

It is a common practice to non-dimensionalise the air mass flows entering the intake by comparing them with the mass flow that would pass through a characteristic area if it were placed in the freestream at flight conditions. Observation of the schematic in Figure 3.5, together with consideration of the simple relationship below will enable the reader to understand the principle leading to the definition of $\varepsilon_d$.

$$\varepsilon_d = \frac{\dot{m}_a}{\dot{m}_{in}} = \frac{A_a}{A_i}$$ (5)

If necessary, this definition can be modified to allow for an internal boundary layer bleed flow.

3.4.1 The Pitot Intake

A form of pitot intake can also be used for supersonic propulsion systems. In this case, however, the intake will require a sharp lip to allow the shock to attach to the front. This will substantially reduce intake drag compared with that of the subsonic round-edged type operating at the same Mach number. Sharp edges, however, do unfortunately result in poor pressure recovery at low Mach numbers. But this design does present a very simple solution for supersonic intake.
If the operating modes of the intake are now examined, firstly consider such an intake operating at supersonic speed and at the critical condition. If the mass flow demands of the engine are reduced, the intake will operate subcritically. Flow no longer required will have to be diverted or spilled over the intake lip. This is impossible if the flow at the lip is supersonic. The normal shock must therefore detach to allow a subsonic spill flow to be set up. As engine demand is reduced and spillage is increased, the shockwave moves forward away from the intake lip. During this stage the shock losses remain constant while the diffuser losses decrease owing to the reducing duct Mach number, resulting in an increased pressure recovery.

It should be noted that for a ramjet at a fixed flight Mach number reduced mass flow demands could arise from (a) a reduction in nozzle throat cross-section area, or (b) an increase in fuel-air ratio.

If the “back pressure” from the engine is reduced by the opposite of one of the two techniques just mentioned, a correspondingly reduced recovery from the intake is achieved by the shock entering the divergent section of the intake. Supersonic acceleration takes place in the inlet duct so the shock occurs at a higher Mach number and so gives a reduced total pressure recovery. The diffuser losses also increase because of increasing shock/boundary layer interaction (flow separation) in the duct.
It should also be noted that if a critical Pitot intake feeding a ramjet operating at a constant fuel/air ratio is subjected to an increased flight Mach number, it will move into a supercritical operating condition.

Since the performance of a Pitot intake is limited by the total pressure ratio across the normal shock, acceptable performance is obtained up to low supersonic speeds, say Mach 1.5. Finally, it should be understood that the pitot intake is stable and shows no hysteresis over the whole Mach number and mass flow range.

3.4.2 The Internal Contraction Intake

In view of the limited maximum freestream Mach number of the pitot intake, it is reasonable to suggest that a frictionless convergent-divergent nozzle operating with reversed flow could form the basis of an intake. While internal contraction intakes can be designed for a single operating freestream Mach number, difficulties are encountered off design.
Assume that isentropic flow has been established in an internal contraction intake duct whose throat area is $A_t$. This is illustrated in Figure 3.7 (a). Then imagine that the throat area is decreased while the inlet area and freestream (i.e., approach) Mach number remain constant. The reduction of mass flow corresponding to the smaller throat area can be achieved only by means of subsonic spillage around the cowl lip. In order that this can happen, a detached shock must be established upstream of the intake. This is illustrated in Figure 3.7 (b). The intake duct will now be operating as a subsonic accelerating device with a consequent loss in performance.

Downstream of the shock the total pressure in the intake duct $p_{od}$ will be less than the value $p_{oa}$ upstream of the shock - the freestream total pressure. Now suppose that the throat area of the duct is gradually increased again, to approach zero spillage. The velocity in the throat is sonic (to achieve maximum value of $X$) so...
and the normal shock is attached to the intake lip. The flow between the intake lip and the throat is assumed to be isentropic. Since \( p_{od} < p_{oa} \), the throat area \( A_t \) will have to be enlarged to equal:

\[
\frac{\dot{m}\sqrt{T_0}}{A_t p_{oa}} = X(\gamma, 1) = \frac{\dot{m}\sqrt{T_0}}{A_t p_{oa}};
\]

for the same flow as originally. When the throat has been enlarged to \( A_t' \) to accommodate the full flow, the normal shock at the lip no longer has a function to perform and is “swallowed” to lie downstream of the throat as shown in Figure 3.7 (c). It is thus seen that at each inlet Mach number there are two flow regimes for the same mass flow. The lower of the 2 curves in Figure 3.8 corresponds to the isentropic case and is achieved by decreasing the throat area causing the internal shock to move upstream from a larger value to the precise value to enable the internal shock to move to the throat where it has “zero strength”. This line is labelled “unstarting limit” because any further reduction in \( A_t \) would cause the intake to unstart, i.e. a detached shock would form upstream of the cowl lip. The upper of the 2 curves in Figure 3.8 corresponds to the precise condition achieved on enlarging the throat to eliminate spill, enabling the shock at the lip to be swallowed. Since this represents the transition from unstarted to started operation, the upper line is labelled “starting limit”. In started operation supersonic internal compression is achieved, there is no spill flow and so low external drag, and good pressure recovery is possible. Maximum internal performance is achieved in practice when the normal shockwave is situated just downstream of the throat and the throat Mach number is just supersonic. In unstarted operation, supersonic internal compression is not achieved and high drag and low performance result.

In Figure 3.8 the vertical broken line with the upwards-pointing arrows represents the process from unstarted operation through starting limit (line B) to started operation, viz: (b) \( \rightarrow \) (c). The vertical line with the downwards-pointing arrows represents the process from started operation through isentropic (line A) to unstarted, viz (c) \( \rightarrow \) (a) \( \rightarrow \) (b).
The operation of a fixed geometry internal contraction intake over a range of flight Mach numbers can now be deduced from the graph in Figure 3.8. It would lie on the horizontal chain dotted lines. At low flight speeds, a detached shock will be formed ahead of the intake. It will be necessary to increase the flight speed up to that given by the higher Mach number curve before the shock can be swallowed. The flight speed must then be reduced close to the value given by the lower curve to establish the normal shock just downstream of the throat and so achieve maximum internal performance. This type of intake thus exhibits a hysteresis effect.

Such a flight sequence is not acceptable and in practice an internal contraction intake must be provided with a variable throat area. To date the mechanical and control problems associated with this proposal have rendered it impractical for missile propulsion.

Perforated internal contraction intakes allowing excess air to bleed from the supersonic diffuser and so obtain automatic control have been suggested. Some air is lost, however, through the perforations at design speed.

NB. In this discussion any boundary layer effects have been neglected.

3.4.3 The External Compression Intake

It will be seen from Gas Dynamics notes, that the total pressure loss across an oblique shock is less than that across a normal shock at the same entry Mach number. This therefore provides a
means of improving the Pitot intake. Suppose, for example, that the free stream entry Mach number is 2.0. Let this flow be deflected (through 15°) by means of a wedge. Then from Gas Dynamics Figure 5.2 the Mach number at exit from the oblique shock so formed would be 1.44 and the shock inclination 45.5°. From the chart on Gas Dynamics, Figure 5.4, the total pressure ratio \( p_{o2}/p_{o1} \) would be 0.95. If this stream were then further decelerated by means of a normal shock, the graphs on Gas Dynamics, Figures 4.4-4.6, give the exit subsonic Mach Number as 0.72 and the total pressure ratio for the normal shock as 0.948. The overall total pressure ratio of the combined oblique and normal shock system would be 0.95 \times 0.948 = 0.90. This compares with a total pressure ratio of 0.72 obtained from a single normal shock from a Mach number of 2.0. A rudimentary form of intake to achieve this is shown Figure 3.9 (a) below, and a practical form in which the subsonic stream is turned back into the flight direction at Figure 3.9 (b). Because some compression takes place ahead of the lip, this type of intake is referred to as the external compression intake.

Figure 3.9 – Schematic showing processes in an external compression intake

By the same token that an oblique shock plus a normal shock were more effective than a single normal shock, two oblique shocks plus a normal shock are even more effective. The single wedge illustrated above may be replaced by double or triple wedges as below. The pressure recovery obtainable from multi-two-dimensional wedge intakes is given in the left-hand graph in Figure 3.10. The recovery from a single normal shock, i.e. that obtained from a Pitot intake, is also plotted for comparison.
It will be noted that the greater the number of shocks the closer the process approaches isentropic, i.e. $p_{od} = p_{os}$. If the number of shocks is made very large, so that the flow deflection is achieved by a series of infinitesimally small deflections generated from a concave surface, isentropic external compression can be achieved (see Figure 3.10). It should be noted, however, that the multi-shock and isentropic intakes deflect the air through successively greater angles. Some losses may be incurred in re-directing the subsonic stream into the axial direction, which detracts from the more efficient systems. Furthermore, the very high cowl lip angle may introduce additional external drag.

In the axisymmetric intake, the two-dimensional wedge referred to above is replaced by a conical centre body; this intake is sometimes called the centre-body intake. It will be appreciated that the flow in a centre-body intake is not two-dimensional. The general qualitative nature of
the flow is the same in both cases, but quantitative differences exist. The performance of an intake having a single conical centre-body of 30° included cone angle is given on the right hand graph in Figure 3.10 for comparison with the two-dimensional intake.

External compression introduces an important factor which can influence the performance of the intake. By virtue of the compression through the oblique shock, pressure forces act on the bounding stream-line behind the shock and before the intake lip (see Figure 3.11).

![Figure 3.11 – Effect of oblique shock position on capture area](image)

Applying momentum theory to the stream tube which enters the intake gives a force in the opposite direction to the flight, i.e. a *drag*. The sum of the pressure forces is known as the *pre-entry drag*. Pre-entry drag can be eliminated by arranging for the shock to impinge on the intake lip, but at the expense of pressure recovery.

Turning to the off-design performance, the behaviour of an *external compression intake* over the range of flight speed and engine back pressure can be deduced by remembering that, in general, the flight speed controls the position of the oblique shock and the engine back pressure controls the position of the normal shock. Consider first the effect of flight speed. The position of the oblique shock only of a single cone or wedge intake is shown in Figure 3.12. If the shock lies outside the lip, the capture area of the intake will be controlled by the shock angle, i.e. by Mach number, but once the shock impinges on the lip or enters the duct, the capture area will remain constant, as illustrated. Additionally, as the Mach number is increased the pressure recovery is decreased, due to the falling total pressure ratio across the shock and the higher leaving Mach number from the oblique shock.
As in the case of the Pitot intake, the normal shock can be located in three positions. In order to obtain maximum recovery the normal shock must be located at the lip, i.e. in the critical position as shown in Figure 3.13:

When the back pressure at the subsonic diffuser exit demands a lower recovery from the intake, the normal shock enters the diffuser. Excess pressure is then lost through the stronger shock. This is the supercritical condition and is illustrated in Figure 3.14:
If less flow is demanded of the intake, the normal shock is expelled to allow subsonic spill over the lip. This *sub-critical* regime is illustrated in Figure 3.15. It should be noted that, because of the reduced internal flow, there will be a reduction in momentum drag but, because of the subsonic flow and increase in cowl pressure, there will also be an increase in engine external drag.

![Diagram showing normal shock subcritical position for an external compression intake](image)

Figure 3.15 – Normal shock subcritical position for an external compression intake

The location of the normal shock and its effect on pressure recovery and capture area are outlined in Figure 3.16.

The pressure recovery obtained from the shock system in the sub-critical region is independent of the mass flow, as indicated by the full line in Figure 3.16. However, excessive spillage can cause pressure losses in the region of the cowl lip, so that the actual curve sometimes takes the form shown by the dotted line. This is particularly so at high flight Mach numbers.
Spillage can be reduced by the introduction of bleed doors in the diffuser section of the intake as shown in Figure 3.17. This will then allow the intake to run nearer to critical condition, thus reducing intake drag and spillage pressure losses. This will be at the expense of a more complex intake design with its associated control system and higher airframe drag due to the ejection of the bleed flow from the intake into the freestream.

When the normal shock is expelled from the cowl it interacts with the oblique shock from the centre body. It will be recalled from the Gas Dynamics notes that a surface of velocity discontinuity (a vortex sheet) originates at the intersection and passes downstream. If this sheet of low-energy air enters the diffuser the effective flow area is reduced, leading to choking of the diffuser and a form of instability known as “buzz”, a rapid oscillation of the inlet shock and flow pattern, which will be discussed in more detail later in these notes. The type of flow pattern giving rise to this instability is shown diagrammatically in Figure 3.18 and the expected pressure
fluctuations are illustrated in Figure 3.19:

Figure 3.18 – Vortex sheet interaction with intake area – “Buzz”

Figure 3.19 – Illustration of buzz characteristics

It would seem reasonable to expect that as the normal shock is expelled further from the cowl lip, and the spillage increased, the vortex sheet would pass well clear of the cowl and stable operation would be restored. Thus, as the flow is reduced, the intake would enter a region of instability and then become stable again as the flow is further reduced. This phenomenon has
been observed experimentally. However, with axisymmetric intakes there is a rise of pressure on the centre body surface, often leading to flow separation. This can generate secondary shocks which intersect the expelled normal shock and give rise to additional vortex sheets. If these approach the cowl lip, as shown in the Figure 3.20 below, an instability can occur as before. This type of flow pattern is more likely to occur at the higher flight Mach numbers than at mildly supersonic speeds.

![Figure 3.20 – High Mach No vortex sheet interaction with intake area – “Buzz”](image)

It should be recorded that knowledge of intake stability is sketchy - most work on intakes having concentrated on maximising the pressure recovery in the stable operating range and simply defining the instability limits. The performance of a typical centre-body intake is summarised in Figure 3.21. The shock configuration at some operating points is indicated diagrammatically.

It should also be appreciated that the preceding treatment is an idealised flow model. In practice, boundary layer build-up, boundary layer/shock interaction, effects of incidence, etc make quantitative estimates over the whole operating range extremely difficult. The effects of incidence, in particular, have proved troublesome in the past. This is not so much an intake problem as such, but that the compressor in the case of the turbojet and the combustion chamber in the case of the ramjet, have been presented with substantial variations in air flow which have led to malfunctioning. The practical success of an intake depends on achieving the quantitative results that are demanded so that the intake can be matched to other components at all flight conditions.
Figure 3.21 – Summary of operating regimes of an external compression intake
4. THE IDEAL THERMODYNAMIC CYCLE

The air breather engine produces work for propulsion using air as the main working fluid. At a selected point in the cycle, heat is added from the combustion of hydrocarbon fuels. In a typical air breather engine the working fluid is not contained in the engine for the complete cycle and operates in what is termed an open cycle where the working fluid enters and leaves the system at a constant rate. In order to analyse the basic thermodynamics of air breather, however, a closed cycle is used to approximate the actual open engine cycle. This approximation is called the air standard cycle, which is based on the following assumptions:

i. A fixed mass of air is the working fluid throughout the entire cycle, and the air is always an ideal gas. Hence $n = \gamma$. Thus there is no inlet or exhaust process.

ii. A heat transfer process from an external source replaces the combustion process.

iii. The cycle is completed by heat transfer to the actual surroundings (in contrast to the exhaust and intake process of an actual engine).

iv. All processes are internally reversible.

v. Air has a constant specific heat.

Although a number of these approximations appear unsuitable, the air standard cycle allows simple qualitative analysis of a number of engine performance parameters such as efficiency and specific work. This cycle, however, is an ideal cycle. Actual air breather engines contain numerous losses which render air-standard ideal cycles inaccurate. These will be discussed in later sections.

4.1 The Brayton Cycle

The air-standard Brayton cycle is an ideal cycle that approximates the gas turbine or ramjet. Figure 4.1 and Figure 4.2 illustrate these cycles. Although both cycles look similar on the T-s diagram, and are based on a constant pressure heat addition and constant pressure heat rejection, the gas turbine cycle has additional processes and has a high degree of mechanical complexity when compared to the ramjet cycle. The presence of rotating components in the gas turbine, however, allows the generation of static thrust which the ramjet cannot generate. Each cycle contains the following processes:

**Gas Turbine**

1-2 An isentropic compression (gases enter a compressor)

2-3 A constant pressure heat transfer (fuel continually burns in a combustion chamber)
3-4 An isentropic expansion (combustion gases enter a turbine)
4-1 An isentropic expansion (combustion gases enter a nozzle)
5-1 A constant pressure heat rejection (exhaust gases are passed to atmosphere)

Figure 4.1 – The basic gas turbine ideal cycle

Ramjet
1-2 An isentropic compression (gases enter an intake)
2-3 A constant pressure heat transfer (fuel continually burns in a combustion chamber)
3-4 An isentropic expansion (combustion gases enter a nozzle)
4-2 A constant pressure heat rejection (exhaust gases are passed to atmosphere)

Figure 4.2 – The basic ramjet ideal cycle

4.2 Cycle Efficiency

It can be shown in the simplest case that the thermal efficiency of the Brayton cycle equals:
\[
\eta_{th} = 1 - \frac{Q_L}{Q_H} = 1 - \frac{c_p(T_4 - T_1)}{c_p(T_3 - T_2)} = 1 - \frac{T_1(T_4/T_1 - 1)}{T_2(T_3/T_2 - 1)} \quad (6a)
\]

Since \( T_4/T_1 = T_3/T_2 \) and \( T_2/T_1 = r^{(1-\gamma)/\gamma} = T_3/T_4 \)

where \( r \) is the pressure ratio \( P_2/P_1 = P_3/P_4 \), the cycle efficiency can be given by:

\[
\eta_{th} = 1 - \frac{T_1}{T_2} = 1 - \frac{1}{(P_2/P_1)^{(1-\gamma)/\gamma}} \quad (6b)
\]

\[
\eta_{th} = 1 - \left( \frac{1}{r} \right)^{(1-\gamma)/\gamma} \quad (6c)
\]

The efficiency of the Brayton cycle is therefore a direct function of isentropic pressure ratio and this is plotted in Figure 4.3.

\[\text{Figure 4.3 – Thermal efficiency of the Brayton cycle as a function of pressure ratio}\]

Improvement to the cycle efficiency is hence dominated by changes in pressure ratio, which is a function of the compressor design in a gas turbine and a function of the flight speed and intake design for a ramjet.

4.3 Specific Work

The specific work \( w \) will determine the size of the gas turbine or ramjet for a given power requirement, since it describes the net work output for a given set of inlet, compression and expansion conditions in the cycle. Specific work is both a function of pressure ratio \( r \) and maximum cycle temperature \( T_3 \) and has units of kW per kg/s of air flow. Hence, in the first case, if the gas turbine specific work is examined: \( w = c_{po} (T_3 - T_4) - c_{po} (T_2 - T_1) \),
Figure 4.4 – Specific work characteristics for a Brayton cycle

This can be expressed in terms of the non-dimensional form:

$$\frac{w}{c_{po} T_1} = t \left( 1 - \frac{1}{\kappa^{(\gamma-1)/\gamma}} \right) - \left( \kappa^{(\gamma-1)/\gamma} - 1 \right)$$  \hspace{1cm} (7)

where \( t = T_3/T_1 \), i.e. the ratio between maximum cycle temperature and inlet temperature. Considering \( T_1 \) and \( c_{po} \) are generally constant, the relationship for specific work can be plotted as a function of \( r \) and \( t \), as shown in Figure 4.4.

Figure 4.4 shows there is a strong dependence of specific work \( w \) on \( t \). Thus, to maximise specific work output from the cycle, there must be the greatest temperature difference possible between the inlet at \( T_1 \) and combustion outlet at \( T_3 \). The simplest way to increase \( t \) is to operate the engine at altitude, which will lower \( T_1 \). The second method is to raise the combustion outlet or turbine inlet temperature \( T_3 \). Materials, however, will limit the maximum \( T_3 \) possible due to their melting points and, in the case of a gas turbine, due to centrifugal stresses on turbine components.

The relationship for specific work also exhibits a maximum at increasing pressure ratio with increasing \( t \). Theoretically, this maximum can be found by differentiating equation (7) and equating it to zero. This results in an optimum pressure ratio \( r_{opt} \) for a given ratio \( t \) to achieve maximum specific work \( w \) which equals:

$$r_{opt}^{(\gamma-1)/\gamma} = \sqrt{t} \hspace{1cm} (8)$$

This characteristic is more easily understood by examining the cycle curves in Figure 4.5.
Figure 4.5 – Optimising specific work output

Figure 4.5 shows three examples of Brayton cycles which all have a maximum combustion temperature limit $T_3$. In all cases, the net work available from the cycle will be equal to the area enclosed by the cycle processes 1-2, 2-3, 3-4, 4-1.

In the first case $\eta_{\text{max}}$, a large pressure ratio is used with a small amount of heat addition to achieve the required $T_3$. This results in a high thermal efficiency but a poor specific work output. In a gas turbine this case is easily understood since such a cycle will contain a large compressor which will require a large turbine to drive it, thus leaving a small amount of net work for propulsion.

In the second case $\eta_{\text{min}}$, a small pressure ratio is used with a large amount of heat addition to achieve the required $T_3$. This results in a low thermal efficiency and a poor specific work output. An example of such a system is an Auxiliary Power Unit for starting aircraft which will have a low pressure ratio allowing for a small, compact compressor but which does require high efficiency because it is only used for short periods.

In the final case $w_{\text{max}}$, the maximum specific work is generated from an optimum pressure ratio which is a value of $r$ between the first two cases but which allows significant amounts of heat addition. With this optimum arrangement it will always result in the turbine output temperature equalling the compressor outlet temperature such that: $T_2 = T_4$. 

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In practice this condition will rarely be met in real gas turbine or ramjet operation, due to constantly changing in-flight conditions. It is likely, however, that the engine will be designed to most closely match this optimum condition for straight and level flight, i.e. cruise.

4.4 Propulsive Work

The previous analysis has concentrated on the cycle efficiency and the available work from the Brayton cycle. To simplify the analysis, it was assumed that net work could be measured in terms of shaft power. In the case of the ramjet, turbojet or turbofan, however, jet propulsion is used to generate a thrust which is converted into useful work through movement of the propulsion unit. Therefore the analysis of efficiency for jet propulsion for a Brayton cycle must be approached in a different manner.

With reference to Figure 4.6, applying the first law to a ramjet or turbojet system:

\[
q + h_a + \frac{u_a^2}{2} + gX_a = h_e + \frac{u_e^2}{2} + gX_e + \dot{w}_e
\]

Thus applying mass flow rates:

\[
\dot{m}_f H_e + \dot{m}_e \left[h_a + \frac{u_a^2}{2}\right] = \dot{m}_e \left[h_e + \frac{u_e^2}{2}\right]
\]

The useful propulsive work \(w_p\) from the system comes from the kinetic energy change across the propulsion unit \((u_e^2 - u_a^2)/2\). The enthalpy change \(c_p(T_e - T_a)\) is unusable, therefore leaving:

\[
\dot{w}_p = \dot{m}_e \left(u_e^2 - u_a^2\right)
\]

These kinetic energy changes are more clearly represented in Figure 4.7 which shows the relative distributions of kinetic energy in the cruise phase of the flight. For a ramjet, these
distributions are simple and actually result in an identical value of thermal efficiency $\eta_{th}$, given a pressure ratio $r$ as defined in section 4.1. For a gas turbine, however, these kinetic energy distributions will vary depending on the compression achieved from the intake. Therefore, at take-off a minor contribution will be received from the intake while at cruise this contribution increases due to the ram effect. In either case, the shaft work required to drive the compressor must be matched by the turbine shaft work. The remaining available work in the expansion can be used to either propel the system using a nozzle as in a turbojet, or in the case of a turbofan drive a second lower pressure turbine connected to a fan through a second shaft.

![Figure 4.7 – Propulsion energy conversion in a ramjet and gas turbine](image)

Hence, considering the kinetic energy changes in the system, the actual thermal efficiency of energy conversion from the powerplant $\eta_{th}$ is found from

$$\eta_{th} = \frac{\dot{m}_e \left( u_e^2 - u_a^2 \right)}{\dot{m}_f H_e}$$  \hspace{1cm} (11)$$

As shown in ‘Introduction to Propulsion, however, the overall efficiency $\eta_o$ is found from:

$$\eta_o = \eta_p \times \eta_{th}$$

where the propulsive efficiency $\eta_p$ for an air breather is found from:

$$\eta_p = \frac{u_a}{1/2(u_e + u_a)}$$  \hspace{1cm} (12)$$

Hence:
\[
\eta_o = \frac{\dot{m}_e \left( u_e^2 - u_a^2 \right)/2}{\left( u_e + u_a \right)/2 \ddot{m}_f H_c} = \frac{\dot{m}_e \left( u_e + u_a \right)(u_e - u_a)/2}{\ddot{m}_f H_c} = \frac{\dot{m}_e \left( u_e + u_a \right)u_a}{\ddot{m}_f H_c} = \frac{F u_a}{\ddot{m}_f H_c}
\]  

(13)

This result matches previous definitions of overall efficiency outlined in the “Propulsion Parameters” notes.
5. THE RAMJET ENGINE

5.1 Thermodynamic cycle

The ideal ramjet engine operates on the Joule (or Brayton) cycle whose most important characteristic is constant pressure heat addition. The cycle, as shown on the T-s diagram in Figure 5.1, consists of:

(i) isentropic compression 1-2 in the intake/diffuser at forward flight speed;
(ii) constant pressure heat addition 2-3 (in the combustor in practice)
(iii) isentropic expansion 3-4 in the nozzle to ambient pressure;
(iv) constant pressure heat rejection 4-1.

The actual cycle, in solid line, differs substantially from the theoretical in that the real effects encountered in the flow processes depress the potential performance. In particular:

(i) the compression process in the intake/diffuser is far from isentropic, particularly at the supersonic speeds generally employed in ramjets;
(ii) both the flame stabilisation and heating processes in the combustor are inherently “lossy”, resulting in a decrease in the total pressure of the flowing gas;
(iii) the nozzle process is not particularly poor but the effect of the imperfect processes upstream of the nozzle is to reduce the enthalpy change available in the nozzle for the creation of the jet velocity, as compared with the ideal cycle;
the closing process is accomplished, in effect, by the continuous rejection of the exhaust gases to the atmosphere and ingestion of fresh air from the atmosphere.

The actual engine thus does not operate on a true cycle at all and will operate as an ‘open cycle’.

5.2 Criteria of Performance

The conventional definition of air breather thrust (see Introduction to Propulsion) is

\[ F = \dot{m}_a u_a - \dot{m}_e u_e + \left( p_e - p_a \right) A_e \]

(14a)

which can be altered, using \( \dot{m} = \rho A u \), into:

\[ F = \rho A u_a^2 A_e \left( \mu \frac{u_e}{u_a} - 1 \right) + A_e p_a \left( \frac{p_e}{p_a} - 1 \right) \]

(14b)

where \( \mu = \frac{\dot{m}_e}{\dot{m}_a} = 1 + \frac{m_f}{\dot{m}_a} = 1 + f \)

The specific size of the engine, as implied by \( A_e \), and the ambient pressure can be generalised by converting the thrust to a thrust coefficient, which is defined like a drag coefficient as:

\[ C_F = \frac{F}{\frac{1}{2} \rho A_u^2 A} \]

(15a)

where the choice of reference area \( A \) is arbitrary. Since the engine thrust coefficient is frequently based on the free stream capture area, we get

\[ C_{F_e} = 2 \left( \mu \frac{u_e}{u_a} - 1 \right) + 2 \frac{A_e}{A} \frac{p_e}{p_a} \left( \frac{p_e}{p_a} - 1 \right) \]

(15b)

which can be easily simplified to

\[ C_{F_e} = 2 \left( \mu \frac{u_e}{u_a} - 1 \right) + 2 \frac{A_e}{\gamma M_a^2 A} \left( \frac{p_e}{p_a} - 1 \right) \]

(15c)

For a perfectly expanded nozzle, equation (15c) simplifies to:

\[ C_{F_e} = 2 \left( \mu \frac{u_e}{u_a} - 1 \right) \]

(15d)

since \( p_e = p_a \)

Making use of the compressible flow relations and assuming that gas properties and specific heat ratios are constant throughout the engine, the following equation relating thrust coefficient, overall total pressure ratio, total temperature ratio and flight Mach number can be deduced from equation (15d):
\[
\frac{C_{F,\text{p}}}{\mu} = \frac{2}{M_a} \left( M_a^2 + 2 \gamma - 1 \left( \frac{p_{\infty}}{p_{\infty}} \right)^{\frac{\gamma - 1}{\gamma}} \right)^{\frac{1}{2}}
\] (16)

In section 5.4, the above equation will be referred to again, when its implications as far as ramjet performance is concerned will be discussed at some length.

Ideal ramjet combustion temperatures can be estimated from:

\[
T_s = \frac{f H_c + c_p T_i}{c_p (f + 1)}
\] (17)

where it is assumed that \(c_p\) remains constant and that the combustion efficiency is 100%.

Another form of thrust coefficient that is used is as follows:

\[
C_F = \frac{F}{\text{lip area} \times \text{ambient pressure}} = \frac{F}{A_i \times p_a}
\] (15e)

This coefficient is somewhat simpler and appears to have advantages in the presentation of off-design performance of a single engine.

“Efficiency” criteria, such as specific fuel consumption, (fuel) specific impulse and overall efficiency have been defined in section 4.4 and “Propulsion Parameters” notes.

**Equivalence Ratio** \(E_R\), is often used in place of fuel-air ratio alone when expressing performance figures and is a ratio defined by:

\[
E_R = \frac{\text{fuel-air-ratio}}{\text{stoichiometric fuel-air-ratio}}
\] (18)

5.3 **Calculation of Thermodynamic Performance**

In estimating the thermodynamic performance of ramjets it is normal to assume one-dimensional flow and trace the flow conditions at each plane of the engine using the flow relationships established in “Gas Dynamics”. The analytical methods available differ primarily in the exactness with which the enthalpy of the fluid is evaluated. In each of the four methods about to be mentioned, both the air and combustion products are assumed to behave according to the ideal gas law, \(pV = mRT\).
Method 1: Constant specific heat.

In this method, it is assumed that throughout the engine specific heat is constant such that \( \gamma = 1.4 \), regardless of the temperature or constituents of the fluid. This approach is the easiest but the least accurate.

Method 2: Arbitrary specific heat.

It is assumed that for processes prior to combustion (in the pure air) specific heat is constant giving \( \gamma = 1.4 \) and for combustion and subsequent processes \( c_p \) is a different value giving \( \gamma = 1.33 \) or thereabouts.

Method 3: Average specific heat.

Due account is taken of the range of temperatures expected to be encountered in a process (or component) and of the composition of the working fluid. From tables of thermodynamic data an average value of specific heat and/or \( \gamma \) is extracted and used to calculate temperatures, etc. Improved average values of \( \gamma \) and \( c_p \) can then be fed back into the calculations to obtain improved estimates of temperatures, etc. This iterative technique is rather tedious and suited to computer solution.

Method 4: Enthalpy.

This is the most accurate technique, evaluating enthalpy above some arbitrary datum temperature according to the relationship \( h = c_p dT \). Tables or charts giving enthalpy as a function of temperature and fuel/air ratio are required.

5.4 Design point performance.

In order to estimate ramjet performance at the design point one may carry out a great number of repetitive calculations along the lines of one of the simpler (or more difficult) procedures outlined in section 5.3. The resulting data could then be plotted in a number of performance maps and the effect of variations in flight Mach number, altitude, maximum cycle temperature (or fuel-air ratio), etc. on specific fuel consumption and thrust coefficient (or perhaps specific thrust) could be assessed. This approach is time-consuming and produces data which are not easily interpreted by the non-specialist. There is a need, therefore, to be able to obtain a concise idea of the broad effects that changes in the main parameters have on ramjet performance. One is thus prepared, in the initial stages, to sacrifice accuracy in the predicted performance for an improved general understanding of performance trends.
One such approximate but concise presentation of design-point performance trends which is due to De Marquis Dale Wyatt (High Speed Aerodynamics and Jet Propulsion, Vol XII, Oxford Univ. Press, 1959) is reproduced in Figure 5.2. This diagram is a graphical representation of equation (16) in section 5.2.

![Figure 5.2 – Ramjet design point characteristics](image)

This plot indicates that for a constant total temperature ratio $\frac{T_0}{T_a}$

(a) at subsonic and low supersonic speeds, the thrust coefficient (or specific thrust at one speed)

is very sensitive to losses in total pressure (irreversibilities) through the engine.

(b) at high Mach numbers (say 4 upwards) the sensitivity of $C_{Fa}$ to decreases in total pressure is very much reduced.

The reason for this trend is that at the very high flight Mach numbers the ram pressure ratio generated is enormous (see Gas Dynamic notes) and so irreversibilities, causing a decrease in total pressure of the gas stream, are of much less significance. This is a general trend which must not be allowed to obscure the influence of cycle temperature ratio also. In general, the higher the ram pressure ratio, the higher is the temperature ratio for, say, maximum $C_{Fa}$. Hence the above diagram shows that ramjet “efficiency” is markedly affected by pressure losses of any magnitude at subsonic and low supersonic speeds, but only by very large pressure losses in the higher speed range (Wyatt). This is an important general truth.

Following on from these generalisations it is now useful to be somewhat more specific. Figure 5.3 and Figure 5.4 (due to Wyatt) show ramjet design-point performance over a range of flight Mach numbers and equivalence ratios (see earlier definition). These performance maps were
plotted from data evaluated by Method 4. Various reasonable assumptions of constant values for diffuser efficiency, combustor inlet Mach number etc. were incorporated, as was the assumption of a fully expanded nozzle \( (p_e = p_a) \). The figures plotted are therefore a very fair indication of design-point performance of conventional kerosene-burning ramjets. From this parametric form of presentation of ramjet performance, the following points should be noted:

Figure 5.3 – Ramjet sfc vs \( M_a \) for a range of equivalence ratios \( E_R \)
Figure 5.4 – Ramjet $\eta_0$ vs $C_{F2}$ characteristics for a range of equivalence ratios $E_R$

(a) At any flight Mach number, the thrust coefficient (and thrust) increases with increase in equivalence ratio (i.e. with increase in fuel-air ratio).

(b) For the assumed diffuser efficiency, the overall efficiency of propulsion increases as flight speed increases.

(c) At the lower flight Mach numbers the maximum efficiency is obtained with the lowest values of equivalence ratio, while at the higher Mach numbers the maximum efficiency is obtained with the higher equivalence ratios. Since the higher fuel-air ratios correspond to higher cycle temperature ratios, it is not surprising that they should turn out to be optimum at higher Mach numbers which correspond to higher cycle pressure ratios.

(d) At the lower Mach numbers (say 2 to 3) it is clear that the highest efficiency is obtained at the expense of thrust coefficient, i.e. specific thrust. The implication of this is that an unduly large engine may therefore result. Since an increased engine diameter (for a given thrust) is bound to mean a heavier engine and a decrease in engine thrust/weight ratio, a balance between high efficiency on the one hand and poor thrust-weight ratio (and increased cross-section affecting drag, possibly) on the other, would almost certainly be sought.

(e) At the higher Mach numbers one enjoys the happy situation of much improved thrust coefficient at the best values of efficiency. However, it must be noted that at the highest values of $M_a$ and $E_R$, the performance quoted is least reliable and in practice the greatest difficulties (particularly in combustion) would occur.
(f) Observation of the Figure 5.3 reveals that, for any equivalence ratio, there is a minimum sfc value against flight Mach number rather than a continuous decrease which might be expected (see variation of $\eta_o$ with $M_a$ in Figure 5.3). This peculiarity is due to the small increase in specific fuel consumption at higher than optimum speeds as compared with the flight velocity change. A brief consideration of the relationship and figure below is helpful in establishing this fact. Here, it can be seen that although the thermal efficiency of the system continually rises causing a fall in fuel flow, the thrust will eventually fall quicker than the fuel flow due to momentum drag resulting in an eventual rise in sfc.

\[ \eta_o = \frac{F u_o}{m_j H_c} = \frac{u_o}{H_c \times sfc} \]

(g) Figure 5.3 also illustrates point (c), in that minimum sfc occurs at low values of $E_R$ at low Mach number, shifting to higher Mach number at large values of $E_R$. To conclude this subsection, the overall performance characteristics are reproduced from the “Propulsion Parameters” notes.
Figure 5.5 – Summary of the performance of various propulsion systems

It is a useful concise indication of sfc’s that could be expected from current kerosene-burning ramjet engines at various design flight Mach numbers at a typical flight altitude. The turbojet and rocket graphs serve as a useful reminder of their comparative performance.

5.5 Off-design Performance.

5.5.1 General considerations.

Needless to say, it is not always possible to operate a ramjet engine at its design condition, however desirable that may be, perhaps in terms of maximising thrust per unit cross-section or minimising sfc. Among “minor” variables affecting performance are ambient temperature and ambient pressure and these are easily accounted for. The angle of incidence of the oncoming air to the intake can also affect performance, as illustrated in Figure 5.6. Considering, however, that the missile will be in a cruise phase while under power for the majority of the flight, this variable is also of minor consideration.

The major flight variable to which a ramjet in a missile is likely to be subjected is flight Mach number. While variations of flight Mach number will be kept to a minimum, it is important to understand the effect such variations have on performance. The degree to which performance at operating conditions other than the optimum (i.e. design) is below that at design depends on the extent of controllable variables in the engine.
Figure 5.6 – Effect of angle of incidence on ramjet intake performance

The easiest and most obvious form of control available is that on fuel flow, i.e. in effect, fuel-air ratio $f$. It is desirable to be able also to exercise control over inlet and exhaust nozzle geometries, but the resultant degree of complexity in both the engine and the controls (and the extra cost, weight and the reduced reliability) has meant that the potential benefits of variable geometry have to date been sacrificed in favour of simplicity.

An illustration of how performance changes at off-design conditions is shown in Figure 5.7. This shows both thrust coefficient and thrust and sfc vs. flight Mach number for a fixed geometry engine whose design point is: $M = 2.0; f =$ stoichiometric; mass flow ratio $\varepsilon_d = 1.0$; intake critical. It is assumed that throughout all flight speeds the fuel control system maintains $f$ constant at the stoichiometric value.

Before considering the performance graphs in some detail, it is necessary to establish how diffuser exit Mach number $M_2$ varies as $M_a$ varies about the design value for a fixed fuel-air ratio. Figure 5.8 forms the basis for the following analysis.
Section 1 - Diffuser inlet (or engine intake).

Section 2 - Diffuser exit (or combustor inlet).

Section 3 - Combustor exit (or nozzle inlet).

Since the nozzle is choked,\[ \frac{\dot{m}\sqrt{T_o}}{A_{p_0}} \text{ is constant.} \]

Also, \( \frac{A_i}{A_d} \) is constant for a fixed geometry ramjet. Hence \[ \frac{\dot{m}\sqrt{T_o}}{A_{p_0}} \text{ is constant.} \]

Hence, \( M_3 = \) combustor exit Mach number = constant

Hence
\[
\frac{\dot{m}\sqrt{T_o}}{A_{p_0}} = \frac{\dot{m}\sqrt{T_o}}{A_{p_0}} \times \frac{\dot{m}_3}{\dot{m}_2} \times \frac{p_{o_2}}{p_{o_3}} \times \frac{T_{o_2}}{T_{o_3}} \times \frac{A_3}{A_2}
\]

(19)

On the r.h.s. of equation (19), the first and second factors are constant. The fifth factor is constant also. Variations in factor 3 are related to \( \Delta T_{o23} \) and will be small; factor 3 can therefore be taken to be substantially constant. Hence:

\[
\frac{\dot{m}\sqrt{T_o}}{A_{p_0}} \propto \frac{T_{o2}}{T_{o3}}
\]

Now it can be shown that at a constant value of \( f \) over a restricted range of flight Mach numbers the total temperature \( T_{o3} \) is also substantially constant. Hence:

\[
\frac{\dot{m}\sqrt{T_o}}{A_{p_0}} \propto \sqrt{T_{o2}}
\]

Since \( T_{o2} = T_{o3} \),

\[
\frac{\dot{m}\sqrt{T_o}}{A_{p_0}} \propto \sqrt{T_{o3}}
\]
i.e. the mass flow function at the diffuser exit will therefore increase with flight Mach number. Typical running lines which meet this condition have been superimposed on the form of intake characteristic curves in Figure 3.21 and have been reproduced in Figure 5.9.

5.5.2 Operations with constant $f$.

We are now in a position to analyse the performance curves of our illustrative example. Let us first consider Figure 5.7 for flight Mach numbers greater than the design value.

(i) $M_a > 2.0$. - As $M$ increases, diffuser exit flow function increases (or $M_2$ increases). This nozzle-dictated behaviour necessitates that the intake/diffuser must operate in the supercritical condition. The terminal normal shock moves downstream into the diffuser passage. The pressure recovery decreases rapidly with increase in $M_a$ and together with the decrease in $\frac{T_{o3}}{T_{o2}}$ (N.B. $T_{o3}$ = constant and $T_{o2}$ increases with $M_a$), causes the reduction in $C_{F2}$.

(ii) $M_a < 2.0$ - As $M_a$ reduces the diffuser flow function reduces (because of the nozzle constriction) and the intake/diffuser goes subcritical. The sharp reduction in thrust coefficient is primarily due to the reduction in air flow rate through the unit and the reduction in $M_a$, hence $\frac{p_{o2}}{p_a}$. Because the captured stream tube is now of a smaller cross-section than the intake area, a pre-entry drag force arises. There could also be additional cowl drag because of the spilling air flow. In practice, it’s also quite probable that the intake would then run into the unstable condition known as “buzz” (see later for details).

One should briefly note what beneficial effects could be obtained with a variable geometry nozzle. For speeds greater than design, the nozzle throat could be reduced to maintain critical (or near critical) operation of the diffuser. Such a technique would improve pressure recovery (compared with a fixed-nozzle off-design condition) and hence improve performance compared with a fixed nozzle ramjet at the same flight speed.

For below-design values of $M_a$, the nozzle throat could be enlarged to maintain maximum air flow through the engine and so off-set the effect of reduced airflow as with a fixed nozzle.

Finally, in Figure 5.7, a plot of relative thrust $F$ against $M$ has been produced for an arbitrary cross-section engine by evaluating:

$$F = C_{F2} \times \frac{1}{2} \rho_v u^2 A_2 = C_{F2} \times \frac{\gamma p_a M_a^2 A_2}{2} = \text{constant} \times C_{F2} \times M_a^2$$

Clearly, for $M_a < M_a$ (design), both $C_{F2}$ and $M_a$ are diminishing so causing a sharp reduction.
in $F$. But for values of $M_a > M_a$ (design), the thrust increases despite the fall in $C_{F_2}$ because of the dominant effect of $M_a^2$. Assuming that the performance calculations at the highest values of $M_a$ are still reliable, the thrust finally levels off and shows a downward trend at around $M = 3.5$. It should be pointed out that in practice such a wide spread of flight Mach number is not likely to be a requirement of a fixed-geometry ramjet.

Figure 5.8 – Ramjet off-design performance characteristics with constant $f$

5.5.3 Operation with varying $f$

As an aid to the understanding of the operation of a typical ramjet at different values of $f$ and $M_a$, Figure 5.9 and Figure 5.10, showing $\eta_d$ versus $X_2$ (with lines of constant $f$ superimposed) and sfc versus $C_{F_2}$, are included.

Consider first the sfc vs $C_F$ curves, corresponding to the intake characteristic curves for a typical isolated ramjet engine. “Isolated” means that the engine performance is treated totally independently of the airframe drag that would be properly associated with a real propulsion installation. This approach is rather idealised but it serves to present typical engine performance trends with the minimum of confusion. The matching of propulsive thrust to airframe drag, and the resultant stability considerations, will be examined in a later section. Some important features should be noted:
At any flight Mach number, when the fuel-air ratio is reduced below the level required to
maintain critical operation, the intake goes into *supercritical* operation with a strong normal shock and reduced pressure recovery. The combined effects of the reduced $\eta_d$ and reduced $\frac{T_o^3}{T_o^2}$ cause a rapid reduction in thrust coefficient (and thrust). Note also that for supercritical operation the $sfc$ worsens but at a low rate initially so that slightly supercritical operation does not have a very serious effect on $sfc$.

When the fuel-air ratio is increased above the value required for critical conditions, the intake is forced into *subcritical* operation with reduced airflow (spillage occurs, causing extra skin friction drag), pre-entry drag and the risk of buzz. Thus although the engine operates with a high value of $\frac{T_o^3}{T_o^2}$ it also has to contend with the reduced airflow and an increased drag so that there is only a slight increase in resultant thrust for a large increase in fuel flow. The specific fuel consumption thus increases rapidly from the minimum at critical when the engine is forced into subcritical operation.

Considering now Figure 5.9 showing the graph of $\eta_d$ vs $X_2$, three running lines giving critical intake operation at flight Mach numbers of 2.0, 2.6 and 3.0 are shown. It should be clear why each of these lines corresponds to a different value of $f$ and, moreover, why the highest value of $f$ corresponds to the highest value of $M_a$. It should also be noted that if critical conditions are established at a cruise Mach number of 3.0, the ramjet would operate in the buzz region at lower Mach numbers. If, on the other hand, the intake is designed to be critical at a lower Mach number, the buzz region can be avoided at the expense of pressure recovery at the cruise Mach number by running supercritical. These rather sweeping conclusions do not necessarily apply to all ramjets but are broadly true for ramjets designed to operate at substantially constant fuel-air ratio.
Figure 5.11 – Ramjet mixture vs thrust-drag coefficient characteristics

The restrictions outlined above can be avoided by relaxing the constant fuel-air ratio conditions. Confining attention to the fixed geometry ramjet, suppose that it were designed for critical conditions at a Mach number of 3.0. Then, in order to maintain critical conditions at the lower Mach numbers, the fuel-air ratio would have to be reduced. If it were desired to maintain critical conditions at all flight Mach numbers the fuel-air ratio would have to take the following values:

\[
\begin{array}{c|c|c|c|c}
M_a & 3.0 & 2.8 & 2.6 & 2.4 \\
f & 0.05 & 0.04 & 0.035 & 0.025 \\
\end{array}
\]

A weaker mixture thus reduces the flight Mach number at which the intake runs critical. Below that critical Mach number, the effects of \( f \) on thrust coefficient will depend on the precise design, but the effects will be small compared with the general loss of \( C_{F2} \) with reduced \( M_a \). The subcritical part of the \( C_{F2} \) vs \( M_a \) curve will therefore remain substantially the same for all mixture ratios. Above the critical Mach number, i.e. in the supercritical range, the thrust coefficient will decrease with increasing \( M_a \) in exactly the same way for all fuel-air ratios. The resulting \( C_{F2} \) vs \( M_a \) curves for constant weak and constant rich fuel-air ratios thus take the form shown in Figure 5.11:

5.5.4 Matching engine thrust to missile drag.

This section on off-design performance of ramjets can now be concluded with a brief discussion on matching ramjet thrust to missile drag.

Superimposed on the curves in Figure 5.12 are 2 typical missile drag coefficient curves. The following observations on ramjet/missile matching may be made:

(i) The low-speed intersections give the lowest speed of ramjet sustained flight, but do not satisfy the condition for stability. Quite apart from considerations of satisfactory
operation of a subcritical intake, above this speed the missile will accelerate; below this speed it will decelerate.

(ii) Changes in the drag coefficient curve will affect the nominal lowest speed of ramjet sustained flight. The fuel-air ratio has little effect on this speed.

(iii) The upper speed intersection gives the speed for steady sustained flight (cruise) and is a stable flight speed. Varying the fuel-air ratio has a significant effect on this flight speed.

(iv) While the lowest speed of sustained (though unstable) flight is little affected by bodily raising or lowering the drag coefficient curve, because the normal shape of the drag curve is steep at the intersection, the high speed sustained condition is significantly altered by changes in the thrust or drag curves.

(v) In broad terms, the variation of specific consumption with speed follows the variation of $C_{F_2}$, except that pressure recovery may be expected to have a diminishing influence on consumption as the speed is increased.

![Figure 5.12 – Ramjet drag-thrust matching characteristics](image)

The following concise statement of some control options may assist in the understanding of matching ramjet engine to missile.
Starting at the critical condition:

**Case 1**

Engine controls selected to maintain constant f. If \( M_a \) caused to decrease, \( D > F \) \( \therefore \) continuous deceleration. If \( M_a \) caused to increase, \( D > F \) but this is a stabilising force restoring system to starting condition. \( \therefore \) This is a *partially unstable* configuration.

**Case 2.**

Engine controls designed to maintain critical operation. Then the system is *unstable* when subjected to either a speed increase or a speed decrease.

**Case 3.**

It would seem to be advisable to design the engine for super-critical intake operation at the design condition in order to be able to maintain a *fully stable* operating condition. Such a design would result in a larger engine to propel a given airframe and cruise operation at a reduced efficiency compared with the engine designed to cruise at critical intake conditions.

### 5.5.5 Variations in ambient conditions.

Turning to variations in ambient conditions at any speed, it is readily seen that the *pressure* affects only density and therefore applies equally to both thrust and drag coefficients. The ratio of mass flow \( \dot{m}_a \) to total pressure \( p_o \) remains the same and the matching is unaffected.

Variations in ambient *temperature*, however, are reflected in variations of \( \frac{\dot{m}_a \sqrt{T_o}}{p_o} \) at the diffuser exit. Thus, reducing the ambient temperature is equivalent to increasing the fuel-air ratio (at a given \( M_a \)). Since the ambient temperature drops with altitude, the demands on fuel-air ratio for increasing speed with simultaneously increasing height tend to cancel out for a fixed-geometry ramjet, and use may be made of this to simplify the control systems with some flight plans. In general, mismatching occurs with variations of ambient temperature and flight speed and appropriate adjustments to the fuel-air ratio must be made.

For some applications, the range of operation obtainable from a fixed-geometry ramjet could be inadequate. The only solution then is to vary the geometry of the nozzle or intake (or both) to maintain matching. As mentioned before, this is a considerable mechanical and control complication and adds to the engine weight. It would therefore be avoided, if at all possible, in missile applications. It could prove to be essential if a substantial speed range was required. The following, Figure 5.13 and Figure 5.14, show typical ramjet performance curves, both dimensional and non-dimensional.
Figure 5.13 – Ramjet performance curve for a Bloodhound MkII (Max Thrust, 0° ISA)

Figure 5.14 – Maximum non-dimensional stub thrust for a Mach 2.5 fixed geometry ramjet
6. THE TURBOJET ENGINE

6.1 Thermodynamic Performance

The turbo-jet engine operates on the Brayton cycle. The cycle is plotted on a conventional temperature-entropy diagram for static and forward-speed conditions in Figure 6.1:

![Figure 6.1 – T-s diagram illustrating the Brayton cycle for a turbojet engine at cruise](image)

It consists of:

(i) **Compression** which takes place in the intake (from 1 to 2) at cruise speed and partly in the intake at zero forward speed and then in the compressor (from 2 to 3) at all flight speeds.

(ii) **Combustion** at substantially constant pressure in the combustion chamber (3 to 4).

(iii) **Expansion** which takes place partly in the turbine (from 4 to 5), providing the work necessary to drive the compressor, and partly in the nozzle (from 5 to 6), where the thermal energy is converted into kinetic energy for propulsion purposes.

The performance of the individual components of turbojet engines has been well established. The overall performance of the engine has also been discussed in Introduction to Propulsion and section 4 of these notes. Further specimen calculations are given in section 3.3 of “Gas Turbine Theory” by Cohen, Rogers and Saravanamuttoo, (Longman, any Edition) and results of such calculations using representative values have been given in Figure 6.2. Curves (a) refer to a turbojet at sea level static conditions and curves (b) and (c) to a turbojet at flight speeds of \( M = 0.75 \) and \( M = 2.5 \) at 36,000 feet altitude. They show the effect of compressor temperature rise.
and turbine inlet temperature on the specific fuel consumption and the specific thrust.

The performance figures given represent the “design point” of a series of engines which meet the specified flight conditions. Compressor and turbine efficiencies etc. have been kept constant at their representative peak values. The figures do not, therefore, apply to the same engine operated over a range of flight conditions. Reduced efficiencies due to mismatching of the components at the off-design conditions have then to be taken into account, as described in section 6.3.

The specific thrust has been used as a measure of engine weight. Thus an engine having a specific thrust of 800 might be expected to have half the weight for the same thrust as a similarly constructed engine having a specific thrust of 400 (neglecting any effects arising from the square/cube law). This simple ratio may be offset in practice by higher compressor temperature rises (i.e. more compressor stages) or more complex turbine cooling systems and so on if these have to be incorporated.

A number of deductions can be made from the graphs in Figure 6.2:

(i) At a given value of the turbine inlet temperature, there is one compressor temperature rise which will give maximum specific thrust, and another which will give minimum specific consumption. The optimum temperature rise will have to be chosen to satisfy the flight plan. It will be noted that optimum temperature rise varies with flight speed and altitude.

(ii) At a given value of the compressor temperature rise there is an optimum value of the turbine inlet temperature, giving minimum specific consumption. The optimum turbine inlet temperature increases as the flight speed is increased.

(iii) In general, increasing the turbine inlet temperature above this optimum increases the specific thrust (i.e. reduces engine weight for a given thrust) but at the expense of specific consumption, unless accompanied by an increase in compressor temperature rise. This is due to the higher jet velocity, and hence reduced propulsive efficiency associated with the higher turbine exit temperatures.

(iv) In general, increasing the compressor temperature rise reduces the specific consumption. The extent to which the specific consumption is reduced becomes less at the higher flight Mach numbers due to the increased importance of the ram effect.
Figure 6.2 – Thrust – sfc characteristics for a turbojet engine
Figure 6.3 – Reheat characteristics of a turbojet engine

One might expect missiles to have shorter flight times than conventional aircraft. In this case greater emphasis could be placed on specific thrust than on specific consumption. A significant limiting factor in this instance is the turbine inlet temperature which is controlled by turbine blade materials and is discussed more fully in section 6.2. Reheat or after-burning is not subject to this limitation and enables more thrust to be obtained from the same frontal area with only a moderate increase of weight, as shown in Figure 6.3. The thrust is obtained by increasing the temperature and hence the velocity and efflux momentum of the exhaust gases. Reheat does not affect the momentum drag and therefore gives an effect which increases with forward speed. However, since the propulsive efficiency is decreased, the extra thrust is obtained at the expense of fuel consumption.

While the use of reheat is not unusual in military aircraft engines, to date the technique has not been employed in a missile engine (in NATO) so far as the author is aware.

6.2 Mechanical Construction.

The mechanical layout of a typical simple turbojet engine suitable for a missile application is shown on in Figure 6.4. It is not possible here to describe the sophisticated design techniques that have been developed for these engines. It must be appreciated that the rotating parts have to be stressed for high centrifugal loads, that fine clearances between the rotating and stationary
parts have to be maintained at all speeds and temperatures, that thermal stresses have to be minimised by kinematic design or by adequate cooling, that bearings have to be properly located, lubricated and cooled, that vibration originating from the rotor has to be accommodated, and that all this must be achieved without significant loss of performance, weight or reliability. It follows that a turbojet is not a cheap engine.

![Figure 6.4 – A turbojet engine suitable for missile propulsion](image)

6.3 Off-Design Performance

Sections 6.1 and 6.2 have been concerned essentially with the design point performance, i.e. the performance at the steady speed and altitude for which all components are most suited. However, the engine may be operated over a range of conditions, particularly if the engine is required to give thrust from zero to full forward speeds at differing altitudes.

The number of variables involved is large and thus requires dimensional analysis to provide parameters which control the performance. Taking the general case, the significant variables involved can be listed:

(i) Independent variables:
   - Air entry conditions $P_a$ and $T_a$
   - Air Mass Flow $\dot{m}$
   - Engine rotational speed $N$
   - Characteristic dimension $D$

(ii) Dependent variables:
   - Thrust $F$
   - Maximum gas temperature $T_{\text{max}}$
   - Fuel consumption $\dot{m}_f$
   - Pressure ratio $r$
   - Overall Efficiency $\eta$

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Now, since there are five independent variables and the three fundamental dimensions of mass, length and time are involved, it is possible to form two dimensionless parameters which contain all the independent variables and express all the dependent variables (in dimensionless forms) in terms of these two parameters.

The relevant independent parameters are: \( \frac{\dot{m}\sqrt{T_a}}{p_a D^2}, \frac{N}{\sqrt{T_a}} \)

The dependent variables can be formed into the following parameters:

\[
\frac{F}{D^2 p_a}, \frac{T_{\text{max}}}{T_a}, \frac{\dot{m}_f}{p_a \sqrt{T_a}}, r, \eta
\]

For a given engine, D will be constant and quasi non-dimensional parameters can be used. Hence:

\[
\frac{F}{p_a}, \frac{T_{\text{max}}}{T_a}, \frac{\dot{m}_f}{p_a \sqrt{T_a}}, \dot{r}, \eta = f \left\{ \frac{\dot{m}\sqrt{T_a}}{p_a}, \frac{N}{\sqrt{T_a}} \right\}
\]

Similarly, using another set of different independent variables, the following could be established:

\[
\frac{F}{p_a}, \frac{\dot{m}\sqrt{T_a}}{p_a}, \frac{\dot{m}_f}{p_a \sqrt{T_a}} = f_i \left\{ \frac{N}{\sqrt{T_a}}, \frac{u_a}{\sqrt{T_a}} \right\}
\]

The latter version is often adopted in engine design brochures as it is more easily applied to specific calculations. The first version is easier to use when only a qualitative understanding of engine off-design behaviour is required. The remainder of this sub-section is devoted to an examination of the off-design performance using these parameters.

(i) A typical set of compressor characteristics over the full mass flow and speed range are shown in Figure 6.5. They have been plotted on the standard non-dimensional basis. The whole range of operating conditions is therefore represented.
Points to note about these characteristics are:-

(a) Each constant non-dimensional speed line has an upper mass flow function limit, corresponding to \textit{choking} in one of the blade rows.

(b) Each constant non-dimensional speed line has a lower mass flow function limit, corresponding to \textit{surge}. Surge is an aerodynamic instability, through which it is not possible to operate the compressor, partly because it would cause mechanical failure of the compressor itself, and partly because combustion would not be possible in an engine due to the unsteady flow.

(c) The line connecting the surge points is known as the \textit{surge line}. It usually has a “kink” in it at about 3/4 design speed due to redistribution of the load between the various stages of the compressor. The kink can be removed by using adjustable blades in the appropriate stages.

(d) The region of high efficiency is confined to a relatively small area around the design point.

(ii) A typical set of turbine characteristics has also been plotted on the same non-dimensional basis in Figure 6.6.
Figure 6.6 – Typical turbojet turbine characteristics

The efficiency of the turbine can be taken as constant over the practical working range. With a fair degree of approximation, a single line can be drawn through all constant speed lines giving a pressure ratio-flow function curve which is independent of speed. Multi-stage turbines follow the empirical “ellipse” law

\[ \frac{\dot{m}}{p_0} \sqrt{\frac{T_o}{p_o}} = k \sqrt{1 - \frac{1}{r^2}} \]  

where \( k \) is some constant determined by the design conditions.

(iii) For equilibrium running of the turbo-compressor it is necessary that:

(a) Mass flow through turbine = Mass flow through compressor

(b) Power provided by turbine = Power to drive compressor

(c) Turbine speed = Compressor speed.

If losses in the combustion system etc. can be neglected, the pressure at the inlet to the turbine can be put equal to the pressure ratio times the pressure at entry to the compressor, \( (= r \times p_{o1}) \). Substituting in eq. (?a) and rewriting
\[
\frac{m \sqrt{T_o}}{p_{o1}} = k \sqrt{\frac{T_{o1}}{T_{max}}} \sqrt{\left( r^2 - 1 \right)}
\]  

For given values of \((\text{turbine inlet temperature}) / (\text{compressor inlet temperature})\) the turbine characteristics can be related to the compressor characteristics as shown in Figure 6.7:

(iv) The turbojet engine characteristics are then determined by the condition for choking in the throat of the nozzle.

\[
\frac{m \sqrt{T_{o5}}}{A_s p_{o5}} = X(\gamma, 1) = \text{constant}
\]

A typical equilibrium running line for the compressor - turbine - nozzle system is illustrated in Figure 6.8:

Figure 6.7 – Characteristics showing matching of turbine and compressor
(a) Increasing the turbine inlet temperature ($T_{\text{max}}$) moves the operating point at constant $N\sqrt{T_{\text{oa}}}$ to a higher pressure ratio and towards surge. This is achieved in practice by using a smaller nozzle, in such a way that condition Error! Reference source not found. is satisfied. Likewise fitting a larger nozzle will result in operation at a lower temperature and a lower pressure ratio.

(b) It is necessary to choose the design operating point so that the highest practical pressure ratio is being achieved, and the compressor is working near its maximum efficiency.

(c) With the selected design operating point the running line should not cross the surge line, and must allow adequate surge margin for acceleration, particularly in the region of the kink.

(d) The actual speed ($N$) of the engine will be fixed by the mechanical design (i.e. stress due to centrifugal forces) and will be independent of the flight conditions. For operation at low forward speeds and high altitude the non-dimensional speed $N\sqrt{T_{\text{oa}}}$ will be high due to the low ambient temperature. There is then a danger of a high speed surge.

(e) When operating at high forward speeds the non-dimensional engine speed $N\sqrt{T_{\text{oa}}}$ will be low due to the high $T_{\text{oa}}$ from the ram effect. The engine may be operating close to
the kink in surge line. This could result in poor handling due to inadequate surge margin, and make the engine liable to malfunctioning. Alternatively some take-off performance may be sacrificed.

(f) When reheat is used the nozzle area has to be changed to satisfy condition Error! Reference source not found. where \( T_o \) is now the reheat temperature. Two position nozzles are often used in practice. Fixed nozzles cannot be used; if the area is correct without reheat it will be too small when reheat is turned on and will throw the engine into surge. If the nozzle area is correct with reheat it will run without reheat, but at a very reduced performance.

(g) At high, particularly supersonic, flight speeds the engine has also to be matched to the intake. At these flight speeds the non-dimensional engine speed \( N\sqrt{T_{oa}} \) will be reduced due to the high ram temperature. It follows from the compressor characteristics (Figure 6.8) that the non-dimensional mass flow \( \dot{m}\sqrt{T_{oa}}/p_{oa} \) must also be reduced, i.e. plotting the non-dimensional flow against flight Mach number would give the curve shown in Figure 6.9:

![Figure 6.9 – Turbojet flight speed – mass flow characteristic](image)

Ideally the intake should operate at critical conditions, which implies (see Figure 3.21) a substantially constant non-dimensional mass flow. The variation of flow with speed given on the previous page does not conform with this condition and the intake must operate non-critically at some speeds. A fixed intake may be designed for critical conditions at any speed. If it is designed to operate in the critical condition at the
supersonic cruise condition (as in Figure 6.10) the low speed variation results in unacceptable pressure recovery at sonic and slightly supersonic flight speeds. If it is designed to be critical near the sonic flight speed (as in Figure 6.11) the running line will enter the sub-critical region at high flight speeds resulting in the danger of intake surge. Some compromise is possible for flight at low supersonic speeds. At high speeds a variable geometry intake or engine, permitting matching at all speeds is necessary. Alternatively auxiliary intakes and bleed systems, depending on the matching speed chosen can be used. All these systems add to the weight and present an additional control problem.

Figure 6.10 – Turbojet intake – compressor matching characteristic at high Mach No
Figure 6.11 – Turbojet intake – compressor matching characteristic at low Mach No

(h) The nozzle has also to be matched to the system. At supersonic speeds a simple convergent nozzle can be used since the nozzle pressure ratio is likely to exceed the critical value only slightly. This also simplifies adjustment of the throat area when reheat is used. However at the higher flight speeds a greater pressure ratio is available across the nozzle (see Figure 6.1). The conditions for maximum thrust (see Introduction to Propulsion section 2.4) cannot be satisfied without the use of a convergent-divergent nozzle. The extra thrust thus obtained over that of a simple convergent nozzle is given below:

<table>
<thead>
<tr>
<th>Flight Mach Number</th>
<th>1.5</th>
<th>2.0</th>
<th>2.5</th>
</tr>
</thead>
<tbody>
<tr>
<td>Increase of Net Thrust</td>
<td>10%</td>
<td>25%</td>
<td>45%</td>
</tr>
</tbody>
</table>

A convergent-divergent nozzle is thus essential at the higher Mach numbers. Unfortunately a fixed convergent-divergent nozzle designed for a Mach Number of 2.0 suffers a 10% - 20% reduction in sea level static thrust due to over expansion. It may also exhibit instability. A variable throat/exit area ratio is therefore necessary, and if reheat is to be used this must be accompanied by a variable throat area. Such a nozzle is difficult to construct. It adds considerably to the weight and introduces a further control complexity.
As a result of the complications of the control system and the additional weight of the fully installed engine it could be concluded that a turbojet could only be used for supersonic applications in which the fuel consumption was of prime importance. This would seem to limit the turbo-jet to aircraft applications. However at subsonic speeds the specific consumption and the simple and light construction makes the turbojet very attractive. If the flight speed was only slightly supersonic so that the penalties of (g) and (h) were small and could be accepted, the turbojet could still appear as a contender in missile applications. The cost per Newton of thrust must however be greater than the figure for a ramjet. The turbojet is thus of limited application for missiles.

Reproduced in Figure 6.12 are the estimated performance characteristics of an up-to-date missile turbojet engine, viz the TELEDYNE CAE 3-402-CA-400 installed in the Harpoon missile.
Figure 6.12 – Characteristics of a commercial turbojet suitable for missile propulsion

The reader may refer to section 2.5 in “Introduction to Propulsion” as an aid to interpretation of these curves.
7. THE TURBOFAN ENGINE

An engine designer can always reduce the sfc for a given subsonic cruise velocity and altitude by selecting a turbofan, (bypass turbojet) rather than a turbojet configuration. The reason for this is that the resultant improvement in propulsive efficiency outweighs the slight worsening in thermal efficiency (lower overall pressure ratio). However, for a given thrust in order to counteract an estimated drag at that flight condition, the engine weight increases as bypass ratio is increased and the frontal area and therefore drag also increases. The consequences of choosing a high bypass ratio rather than a moderate one or no bypass at all are therefore clear.

If the effects of increased engine weight are considered, for example, it is reasonable that a possible criterion of performance might be the sum of engine weight and fuel weight for a given range or duration. One would thus be trading off the reduced fuel requirements of the turbofan against its greater weight, seeking the best combination, i.e. that giving smallest overall weight. Such a procedure is a well-established part of the optimisation process.

The turbofan will probably never be widely used in missile propulsion. Only for very long range or great endurance (an extended loiter, perhaps) will the extra weight, complexity and cost of the turbofan be justified (e.g. Tomahawk). Turbofans that are employed in missiles will be for subsonic flight only, and will have modest bypass ratios, typically no greater than about 2. It is in the application to the large, wide-body, long-range subsonic transport aircraft that bypass ratio of the order of 4 to 8 pay off.

Optimisation of the turbofan cycle is rather complex since there are 4 main thermodynamic parameters that have to be considered. With reference to Figure 7.1 these are:

(i) overall pressure ratio (r) (including ram pressure ratio)
(ii) turbine entry temperature (TET) (i.e. maximum cycle temperature)

(iii) bypass ratio $\lambda = \dot{m}_c / \dot{m}_h$ (“cold” flow / “hot” flow)

(iv) fan pressure ratio.

The procedure, which need not be discussed in detail, involves much tedious calculation and results in the following broad truths:

(a) increasing bypass ratio reduces sfc but causes a significant reduction in specific thrust also; hence high bypass ratio engines tend to be fuel efficient but have low thrust/frontal area;

(b) The best fan pressure ratio decreases as bypass ratio increases, while the best fan pressure ratio increases as TET increases.

In the long-range subsonic aircraft the prime need is to minimise sfc in order to maximise range. This requirement is easily deduced from the Breguet range equation (see the “Velocity & Range Estimation” set of notes). Such long-range aircraft tend to employ high pressure ratio and high bypass ratio engines. Provided a significant reduction in sfc is obtained at only a small cost in terms of $L / D$ and $m_o / m_b$, the use of the large bypass ratio can be justified. It should be noted, however, that propulsive performance is not the only consideration.

In missile propulsion, moderate TET, pressure ratio and bypass ratio are likely to be required and for these features the 2-spool arrangement (see Introduction to Propulsion) is adequate. The reasons for arriving at this conclusion are as follows:

(a) Missile diameters are invariably very restricted in order to minimise drag, to reduce radar reflecting area and so to reduce the likelihood of detection and to standardise, and simplify, launch techniques. Since the engine will almost certainly be integral with the missile body, the engine diameter will also be limited, thus tending to favour a small bypass ratio.

(b) Since missile engines are likely to be small, compared with typical aircraft engines, the feasibility of a turbofan is even less likely. The core of such an engine would have to be even further scaled down, thus depressing thermal efficiency and hence increasing specific fuel consumption.

(c) By definition a missile engine is expendable, so there are very definite needs to maintain simplicity and cheapness of design. These features are most likely to be obtained with modest pressure ratio and bypass ratio.

(d) It is doubtful if sfc will ever be the all important characteristic of a missile
propulsion unit, for typical ranges, compared with frontal area, length, weight, complexity, cost, reliability etc, so, again, the simpler engine configuration is favoured.

Figs 7.2 to 7.5 show sets of performance data which illustrate how turbofan engine performance compares with turbojet engine performance.

Figure 7.5 is for a family of engines having overall compressor pressure ratio of 7.5/1, an optimised fan pressure ratio and $T_{\text{max}}$ of 1300K. The take-off condition can be regarded as the design condition and other flight Mach numbers are therefore for part-load or off-design conditions.
Figure 7.4 – Relationship between fan pressure ratio and specific engine thrust

Figure 7.5 – Characteristics of turbofan engine thrust for a range of flight speeds